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THE FLOW ABOUT THE LEADING EDGE OF A SWEPT BLUNT PLATE AT HYPERSONIC SPEEDS

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AERONAUTICAL RESEARCH LABORATORIES
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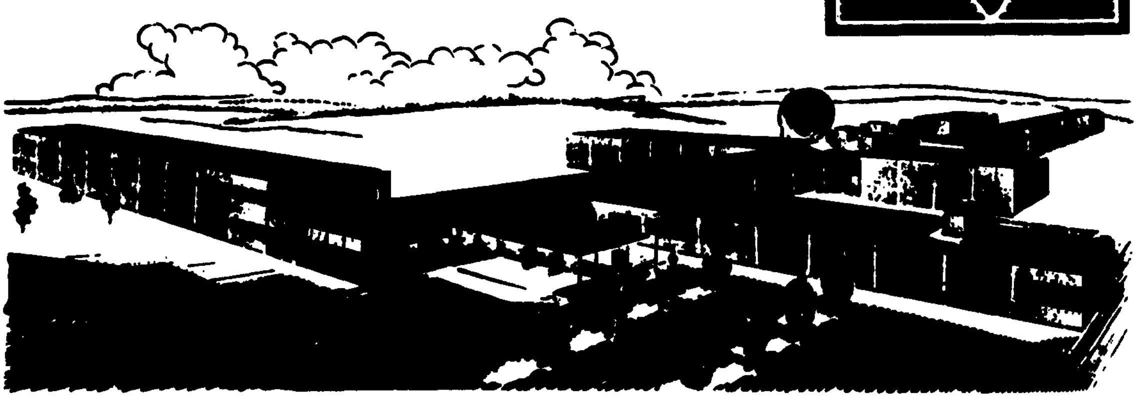
UNITED STATES AIR FORCE



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FOREWORD

The present study is a part of a program of theoretical and experimental research on hypersonic flow being conducted by the Gas Dynamics Laboratory, The James Forrestal Research Center, Princeton University, Princeton, New Jersey on Contract AF 33(616)-7629 for the Aeronautical Research Laboratories, Office of Aerospace Research, United States Air Force. The work reported in this interim report was on Task 7064-01, "Research on Hypersonic Flow Phenomena" of Project 7064, "Aerothermodynamic Investigations in High Speed Flow" under the technical cognizance of Capt. Walter W. Wells of the Hypersonic Research Laboratory of ARL.

ABSTRACT

As part of a fundamental study of hypersonic wings, some detailed pressure distribution and heat transfer results have been obtained over sections of the leading edge region of a blunt plate at sweep angles from zero to 75°. The tests were carried out in the Princeton University Helium Hypersonic Wind Tunnel at Mach numbers from 7 to 17. Some effects of the apex or upstream boundary of the plate on the leading edge regions under study were determined at various stations along the leading edge. The leading edge regions examined showed deviations from normal Mach number considerations at high sweep angles over the entire Mach number range studied.

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SYMBOLS

c	Specific heat of stainless steel, Btu/lb ^{O}R.
h	Heat transfer coefficient, Btu/ft^{2} sec ^{O}R.
M	Mach number
Nu	Nusselt number
p	Pressure, lb/in^{2}
p_l'	Free stream static pressure at an orifice location if no model were present
Pr	Prandtl number
r	Radius of cylinder, inches
Re	Reynolds number $\frac{\rho u x}{\mu}$
s	Distance from the "stagnation" line of the model, inches
t	Leading edge thickness of model, inches
t^*	Time, seconds
T	Temperature, ^{O}R.
u	Velocity, ft/sec
x	Distance from the apex of the model, inches
θ'	Normalized enthalpy function $(H-H_w/H_e-H_w)$
μ	Viscosity, slug/ft sec
ν	Kinematic viscosity, ft^{2}/sec
ρ	Density, lb/ft^{3}
τ	Model skin thickness (heat transfer model), inches
Λ	Sweep back angle
Δ	Standoff distance of shock, inches
du_e/dx	Velocity gradient at the stagnation point at the edge of the boundary layer

SUBSCRIPTS

e	Edge of boundary layer
in	Inside surface of thin skin of model
M	Measured pressure
N	Pressure calculated using the normal Mach number across a normal shock
out	Outside surface of thin skin of model
r	Recovery
w	Conditions on the wall
1	Free stream
0	Free stream stagnation
Λ	Conditions pertaining to yawed cylinder
$\Lambda=0$	Conditions pertaining to unyawed cylinder

INTRODUCTION

As flight velocities increase, the problem of aerodynamic heating of lifting and control surfaces becomes increasingly serious. A method of alleviating this problem is to increase the radius of curvature of the leading edge and increase the sweepback angle. The increased radius of curvature, although increasing the drag of the body, decreases the heat transfer rate at the stagnation point and has been well established by studies on spheres. The increase in sweepback angle decreases the heat transfer by reducing the pressure level over the nose. Although some theories have been developed for a blunt body with sweep, there is serious lack of detailed experimental data in the hypersonic flow regime and for wide Mach number variations. For these reasons, a study of the nose region of a blunt two-dimensional plate at various angles of sweepback has been carried out. The primary purpose of these tests was to examine the portion of a swept wing leading edge which might be considered two-dimensional (if any), i.e. analogous to the infinite swept wing with no effect of the apex. The use of the normal Mach number hypothesis to predict the pressure distributions and heat transfer at various sweepback angles and comparisons with other available theories were to be examined. Since these theories are mainly based on two-dimensional flow considerations, their application to the flow over swept wings is only true when the three-dimensional effects are small and this region was still to be determined.

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The configuration studied had a leading edge which was cylindrical in the direction normal to the leading edge of the model and was tested at zero angle of attack. This enabled easy correlation of the results with theory since the location of the "stagnation line" is known and the flow over a cylindrical rod normal to the flow direction has been extensively examined. Sweepback angles from zero to 75° at Mach numbers from 7 to 17 have been investigated. Some results from the present paper were given during January 1962 at the Institute of the Aerospace Sciences 30th Annual Meeting (Ref. 11).

BRIEF REVIEW OF THEORIES

The theories available for predicting the pressure distribution over swept wing leading edges are limited. Considering the chordwise pressure distribution, "modified Newtonian" has some theoretical justification and seems to fit experimental results at lower Mach numbers over the nose region.

For the effect of sweep, the pressure at the "stagnation line" can be calculated, using the normal Mach number approximation which applies to a supersonic leading edge. This body is considered two-dimensional using the normal Mach number and neglecting the crossflow component. This, of course, assumes a stagnation point. In reality, the neglected crossflow component (along the leading edge) is, for reasonable sweep angles and hypersonic stream Mach numbers, supersonic and there is no "stagnation line" for a swept wing. The term stagnation line will be used herein to identify this region of the model.

Many theories have been developed for predicting the heat transfer over a blunt body. They all include the following basic assumptions, plus a few additional ones:

1. perfect gas
2. constant specific heats
3. isothermal bodies
4. Prandtl number equals a constant.

Theories by Lees (Ref. 1), Eckert and Livinggood (Ref. 2) and others will not be considered due to the additional assumptions of a highly cooled wall, which does not apply for the present case, or difficulties involved with direct correlation of the theory with experimental results (the evaluation of the flow quantities at the edge of the boundary layer).

Reshotko and Beckwith (Ref. 3) considered the effectof sweepback on heat transfer using a linear viscosity temperature law in addition to the above assumptions. The three-dimensional boundary layer equations were transformed to the incompressible region using the Stewartson transformation. To solve these equations, a Falkner-Skan velocity distribution in the transformed coordinates was assumed. This limits the development to stagnation line flow where this type velocity distribution actually occurs. The boundary layer equations were then solved numerically including the effects of arbitrary Prandtl number and non-insulated surfaces. Assuming a modified Newtonian pressure distribution at the stagnation point, the heat transfer variation is represented by

$$\frac{h_\Lambda}{h_{\Lambda=0}} = \left(\frac{\theta'_{w,\Lambda}}{\theta'_{w,\Lambda=0}}\right)_{Pr=1} \left[\cos\Lambda \frac{\left(\frac{p_{w,\Lambda}}{p_l}\right)\left(\frac{D}{u_l}\frac{du_e}{dx}\right)_{M_{ln}}}{\left(\frac{p_{w,\Lambda=0}}{p_l}\right)\left(\frac{D}{u_l}\frac{du_e}{dx}\right)_{M_l}}\right]^{\frac{1}{2}}$$

where $\left(\frac{\theta'_{w,\Lambda}}{\theta'_{w,\Lambda=0}}\right)_{Pr = 1}$

is obtained from the numerical solution, du_e/dx from the modified Newtonian pressure law and p_w from experimental results. It was noted by Reshotko and Beckwith (Ref. 3) that even though the above equation was for a Prandtl number equal to one, the error involved for a Prandtl number of 0.70 was found to be less than two percent.

Goodwin, Creager, and Winkler (Ref. 4) considered the incompressible case with a linear viscosity temperature law in addition to the above assumptions. In the limit for large Mach number they obtained the geometrical result that:

$$\frac{h_\Lambda}{h_{\Lambda=0}} = \cos^{3/2} \Lambda$$

Beckwith and Gallagher (Ref. 5) used the similar solutions obtained by Beckwith (Ref. 6) which included the above assumptions in addition to a linear viscosity temperature relationship and a modified Newtonian pressure law at the stagnation point. They obtained:

$$\frac{h_\Lambda}{h_{\Lambda=0}} = \left(\frac{\mu_w}{\mu_o}\right)^{\frac{1}{2}} \left(\frac{T_o}{T_w}\right)^{\frac{1}{4}} \left(\frac{p_w}{p_{w,\Lambda=0}}\right)^{\frac{1}{2}}$$

which applies to the stagnation line.

At a two-dimensional stagnation point Reshotko and Cohen (Ref. 7) obtained a solution using the results from Cohen and Reshotko (Ref. 8). The assumptions included a linear temperature viscosity law and a Falkner-Skan velocity distribution in the transformed coordinate system in addition to those previously mentioned. The Stewartson transformation was applied to the two-dimensional boundary layer equations and solved numerically for $Nu/\sqrt{Re_w}$. The heat transfer can then be represented by:

$$h = \frac{k_w}{\sqrt{\nu_w}} \quad \frac{Nu}{\sqrt{Re_w}}\left(\frac{du_e}{dx}\right)^{\frac{1}{2}}$$

where du_e/dx is obtained from experimental results.

Reshotko (Ref. 9) further simplified the analysis of Cohen and Reshotko (Ref. 10) for the chordwise distribution of the heat transfer over a two-dimensional body. The approximations include a linear viscosity temperature law, and the ones previously mentioned. Stewartson's transformation is applied to the two-dimensional boundary layer equations and the result is expressed in terms of dimensionless parameters related to the wall shear, the surface heat transfer, and the transformed free stream velocity. Next Thwaites' concept of the unique interdependence of these parameters is assumed. The correlation numbers for these parameters are then calculated using the exact solutions of Ref. 8. Reshotko then makes a few additional assumptions in approximating the correlation numbers for heat transfer in terms of the pressure gradient observed on the body. He arrives at an expression for the heat transfer as a function of the pressure distribution over the body. This can be easily calculated after a simple numerical integration. He states that these results should be within ten percent of the exact values for moderate pressure gradients and/or highly cooled surfaces but proposes no method for application to swept leading edges.

EXPERIMENTAL FACILITIES AND MODELS

The test program was conducted in the three inch diameter, Princeton University Helium Hypersonic Wind Tunnel (Ref. 12). The tests at Mach numbers of 8, 13, and 17, were conducted in conical nozzles which had

Mach number gradients of about 0.4 per inch for the lower Mach number range and 0.7 per inch at the high Mach number range. The tests at a Mach number of 11 were conducted in a contoured nozzle. The free stream Reynolds number varied from 0.5×10^6 per inch to 1.0×10^6 per inch. Helium at room temperature was used as a test fluid.

The pressure model was 1/8 inch thick, 1 inch long and 2 inches wide, and was constructed of steel (Figure 1). The location of the pressure taps is shown in Figure 2. The basic model consisted of a rectangular portion on which the pressures were located. To this, various dummy tips were attached to form a partial delta wing for different sweepback angles (Figure 2). Tips were also constructed to change the distance from the apex. The model was mounted from the side of the tunnel (model in the vertical plane) and rotated about the leading edge.

The heat transfer model was of the thin skin type, made of stainless steel 0.0095 inches thick (Figure 1). The physical dimensions of the basic model were the same as the pressure model with the thermocouples located at the same stations as the pressure orifices. Tips similar to those used with the pressure distribution models were attached to the basic heat transfer section, but were made of lucite to limit heat conduction effects. The thermocouples were copper-constantan wires of 0.010 inches diameter. Holes 0.006 inches in diameter were drilled in the skin 0.016 inches apart. The wires were etched to approximately 0.005 inches in diameter and inserted in the holes. They were then spot welded in place from the outside. The model was mounted in exactly the same manner as the pressure model.

To obtain recovery temperatures a lucite model was constructed to have the same physical dimensions as the heat transfer model. A slotted groove was made 0.002 inches deep on the surface through which the thermocouple wires, 0.003 inches in diameter, were brought. The groove was filled with silver circuit paint and the wire cut level with the surface. These junctions were at the same location as on the heat transfer model. This was done to minimize possible conduction errors which were calculated to be less than 0.2 percent.

The pressures over the models were recorded on manometers using silicone oil and mercury with a 20 micron reference pressure. Copper tubing was used throughout to prevent outgasing.

The heat transfer was measured using the transient technique. The temperature time history of the model was recorded on Leeds and Nothrup Speedomax recording potentiometers having a full scale response of less than 1/4 second. The initial temperature of the model was set higher than room temperature by using heat lamps and then cooled to recovery temperature during the run. The heat lamps were turned off after the test section was evacuated by the ejector system approximately 30 seconds before the run began. This eliminated any heat input due to the slow cut-off time of the lamp, and since the test section was at a relatively low pressure, about 1/10 psia, the temperature varied very little until the beginning of the run. The tunnel was equipped with a quick start mechanism that established the flow in about one millisecond. The temperature time history of each thermocouple was recorded for as much as the first five seconds of the run during which time the stagnation temperature varied only one degree.

To obtain the recovery temperature, the lucite model was used. The tunnel was run until no temperature variation could be detected (until equilibrium was established), then the temperature recorded.

Four dummy models were built for the express purpose of determining the shock wave shape along the leading edge by optical means. Two had a sweepback angle of 60°, one with a blunt apex and the other with a sharp apex; and the other two had a sweepback angle of 75°. To check the general two-dimensional character of the flow, several exploratory oil trace studies were made. These studies were carried out by injecting a very low viscosity oil through two of the pressure orifices on the stagnation line of the model during the test.

DATA REDUCTION

To determine the heat transfer coefficient, radiation and conduction along the model and down the thermocouple wires were neglected. The errors involved were minimized by the thin skin and small temperature differences for the first few seconds of the run. Solution of the one-dimensional heat flow equation, assuming a thin skin, gives:

$$h = \frac{\rho c}{T_w - T_r} \tau \quad \frac{dT}{dt^*}$$

In the nose region, where there is curvature, the two-dimensional heat equation must be considered. Working in polar coordinates the solution is:

$$h = \frac{\rho c}{T_w - T_r} \quad \frac{r_{out}^2 - r_{in}^2}{2r_{out}} \frac{dT}{dt^*}$$

The above analysis is very similar to that given in Ref. 13. For the case of the present model the effective thickness in the nose region is seven percent lower than the actual thickness. From the temperature-time history, the slope at time equal to zero can be found and from this the heat transfer coefficient can be calculated.

The slope at time zero is used since the conduction errors are minimized and T_w is known. The precision of the results at time equals zero is compromised by the finite response times of the recorders and the tunnel starting time. To minimize this effect, the zero time conditions were obtained by extrapolating back to zero time by means of a computer. The method used was as follows: The first 1/4 second of data was neglected. The extrapolation was done by fitting an exponential curve to the data after 1/4 second and working back to zero time. Since pure convective heat transfer is a logarithmic function, a calculation of the quality of the curve fitting to the data was done by the method of least squares. Only curves having a correlation coefficient of 0.99 and better, were used. In the cases of high heat transfer (stagnation point) the first second of recorded data was used for the data reduction. For the low heat transfer rates obtained experimentally on the flat plate region of the body, 4 to 5 seconds of data were employed.

To determine the error involved in the method of obtaining the slope at time zero of the temperature time history, a discharging capacitor circuit was used as an input to the recorders. The electrical circuit was constructed with the capacitance and resistance in parallel, with the input resistance approximately the same as that of the thermocouple junctions used. The capacitance was varied to obtain the entire range of initial slopes obtained experimentally. The initial slopes were then calculated in the same manner as for the heat transfer data and checked to within four percent of the theoretical values calculated for the electrical input circuit. In an attempt to try to estimate the accuracy of the heat transfer values, it was necessary to consider the errors inherent in the model as well as those due to the measurements. The skin thickness of the model was known to ± one percent. The properties of the material, namely the density, specific heat and thermal conductivity were obtained from the best sources available (Ref. 14).

The errors in the thermocouple measurements show up in 2 parts. For the steady state recovery factor, the maximum error of 0.2 percent causes an error of two percent in the heat transfer. The transient temperature measurements from the temperature-time curve produces a four percent maximum error in the initial gradient due to machine response time (as indicated previously). Both these errors tend to lower the values of the measured heat transfer from six percent to about two percent. In calculating the heat transfer, both the conduction along the model surface and radiation from the model were neglected. The radiation error was calculated and found to be less than 0.1 percent. The conduction along the thin skin was less than two percent (due to the fact that the heat transfer coefficient was evaluated at zero time). The conduction losses down the thermocouple wires were estimated to cause a maximum reduction in the heat transfer coefficient of six percent but the data were not corrected for this since relative heat transfers were of primary interest in this study.

DISCUSSION AND RESULTS

A summary of the pressure data is presented in Figures 4, 5, 6 and 7. It should be noted here that $s/t = 10^{-1}$ is actually the stagnation line, $s/t = 0$. The dash lines represent modified Newtonian theory based on normal Mach number. It can be seen that the theory generally agrees with the experimental values up to body angles of approximately 45° and drops below the experimental points after this station. This is due to the fact that the actual pressure must approach the free stream pressure and the theory goes to zero at the shoulder. The differences between theory and experiment at the stagnation point will be discussed in detail in a later section of this paper. Each data point represents several experimental points since the scatter for pressure measurements is negligible. The solid lines represent the flat plate data of Refs. 15 and 16 which were obtained over the downstream flat plate sections of similar models.

To check the effect of the apex shape, both sharp and rounded apexes were tested at Mach numbers of 11 and 17 at sweepback angles of 60^o, 70^o and 75^o. The pressure over the measuring stations (x/t from 10 to 25, approximately) did not vary within the experimental accuracy (order of one percent).

The effect of stagnation pressure was also investigated as the heat transfer tests were performed at 1000 psia and the pressure tests from 1000 to 400 psia. It was found that the variation in the pressure distribution caused by changing the leading edge Reynolds number (based on free stream conditions and leading edge thickness) from 50,000 to 120,000, was only six percent (Figure 8). At the lower stagnation pressure a slightly higher pressure was measured which is to be expected due to the greater viscous effects.

The pressures along the stagnation line were then examined in detail for sweep angles of 60^o and 75^o. At 60^o, the flow was found to be two-dimensional (that is, the pressure along the leading edge was constant, Figure 9). On the graph, p_M represents the measured pressure and p_N the pressure calculated considering the normal Mach number across a normal shock. The fact that the pressure ratio fell below one will be discussed in greater detail later. At 75^o, the flow was found to deviate from two-dimensionality which was also observed by Penland (Ref. 17) at a sweepback angle of 70^o, at a Mach number of

6.8 in air. In the present results, the pressure ratio along the stagnation line initially increases and then decreases to a value less than unity, rising again at the station furthest from the apex. The insensitivity of the results to the apex shape can be noted from a comparison of the few sharp apex results with the major results obtained with the blunt apexes.

To further examine this departure from two-dimensional flow at high sweep angles, some oil trace and optical studies were undertaken. The oil studies at sweepback angles from 37 1/2° to 75° are summarized in Figure 11. A small amount of very thin oil was injected through a stagnation line orifice and the resulting streak was photographed. This technique gives the flow direction at the <u>surface</u> of the model but, of course, does not define the <u>stream</u> direction. The flow direction is from left to right and the leading edge of the model is indicated by the hatched line. The oil streak (flow direction on the surface) is shown by the heavy line marked with arrowheads. Up to about 60° sweep, the flow on the surface over the leading edge is two-dimensional, i.e., perpendicular to the leading edge. The effect of increasing the sweep angle is noticed by the flow deviating from this "two-dimensional" flow closer to the shoulder of the nose section. For sweep angles of 60° and higher, three-dimensional effects are noticed over the entire leading edge region but there seems to be no major change (over the nose region) in changing from 60° to 75° sweep angle.

To supplement the oil trace studies, shadowgraphs were taken normal to the plane of the wing, and the stand-off distance was measured on an optical comparator, (Figures 12 and 13). The accuracy of the measurements made from the shadowgraphs is indicated as a band on the figures. Results were obtained for both blunt and sharp apexes. For the 60° case, the stand-off distance varies as might be expected from simple considerations. Since the stand-off distance is less for a three-dimensional nose than a two-dimensional one of the same dimension, one would expect the stand-off distance to increase from the apex, approaching some constant value when the effect of the apex was small. Both the sharp and blunt tip exhibit this trend, and are approximately constant in the instrumented region. This agrees with the pressure data and the oil trace study. For the 75° case, there is no uniform region obtained over the dimensions of the test model. Although the strong effect of the blunt apex seems to have died out (as compared to the sharp apex), considerable variation in the stand-off distance along the measuring region is noted. This change in the stand-off distance in the region of measurement agrees with the previously mentioned pressure measurements.

A more direct comparison of the normal Mach number hypothesis for variable Mach number is shown in Figure 14. The experimental results on the stagnation line are compared to simple normal Mach number predictions. The leading edge Reynolds number for each case is noted in the figure. Each indicated point represents the average of many data points whose scatter was less than two percent. The hypothesis is seen to be correct only at low sweep angles. At higher sweepback angles, the measured pressures fall above the predicted values for higher Mach number, and below the predicted values for low Mach number. Since the flow at the highest sweepback angles (70° and 75°) was three-dimensional these data should be considered doubtful, but the same effect is clearly noted at sweepback angles of 60° and less where the flow has been shown to be two-dimensional. No mechanism has been proposed to explain this effect and further tests are planned for higher Mach numbers to see if this trend continues.

A summary of the heat transfer data at a Mach number of 11 is presented in Figure 15. For a value of s/t and Λ, each indicated point represents about 20 individual values of h (from either change of spanwise location of the junction, model changes, or different tests). The repeatability of the values for one physical point on a single model was $\pm$ two percent. The experimental scatter for each data point caused by different models, test conditions, data reduction, and variation in the spanwise direction was a maximum of $\pm$ eight percent. The general trend of the data, as expected, agrees with the pressure data. At large s/t the heat transfer approached the flat plate value predicted by theory. The solid line represents the theories by Cohen and Reshotko at zero sweep based on the measured pressure distribution using the temperature external to the boundary layer to evaluate $Nu/\sqrt{Re_w}$ over the entire body. To check the normal Mach number hypothesis, Cohen and Reshotko's theory was extended to the sweep case using the measured pressure distributions. The results of this simplified approach were in poor agreement with the present data being as much as fifty percent in error.

A comparison of the stagnation values of the present results with theories and experiments for various sweepback angles is shown in Fig. 16. Here the heat transfer coefficient is non-dimensionalized by its value at zero sweep. Theories by Beckwith and Reshotko; Beckwith and Gallagher; and Goodwin, et.al. ($\cos^{3/2}\Lambda$); are represented by the solid and dashed lines. Experimental results by Feller (Ref. 18) and Goodwin, et.al. (Ref. 4) obtained in air at lower Mach numbers are also represented. The present data are in good agreement with Goodwin's theoretical result at high sweep but are considerably below the other experimental data. The uptrend of the data at the highest sweep is probably caused by the noted three-dimensional effects. As the effective radius of curvature is smaller (the leading edge viewed from the free stream direction is elliptical) the measured heat transfer would be higher.

Looking at the stagnation line in detail, no major deviation was observed along the span at $\Lambda = 60^{\circ}$ (Fig. 17). This agrees with the pressure and stand-off distance results with the indication that the flow in this nose region can be considered two-dimensional. At $\Lambda = 75^{\circ}$ (Figure 18), the heat transfer rates were somewhat less than that found for the 60° study, but decreased along the model. This again agrees with the pressure results (Figure 15). It is planned to examine this effect in greater detail in continuing studies.

Preliminary results for these conditions presented in an IAS preprint (Ref. 11), which showed much higher heat transfer at the upstream region of the measuring regions were found to be influenced by erroneous recovery temperature measurements which were corrected by these subsequent studies.

CONCLUDING REMARKS

The results which have been obtained from this exploratory study to date follow:

1. The apex shape had no effect on the pressure distribution in the region of measurement for all sweepback angles.

2. Over the nose region, two-dimensional flow was observed over the instrumented section to $\Lambda = 60^{\circ}$.

3. At low sweepback angles, the Mach number had no effect on the pressure distribution as predicted by the normal Mach number hypothesis.

4. Modified Newtonian theory agreed well with the measured pressure in the stagnation region and up to $\theta = 45^{\circ}$.

5. In the Mach number range from 7 to 17, the stagnation point pressure predicted by the normal Mach number hypothesis was in good agreement with the measured values for sweep angles less than 30°. At higher sweep angles, the measured pressure deviated from the predicted value by $\pm$ ten percent over the Mach number range.

6. The stagnation point and chordwise heat transfer distribution at zero sweep as predicted by Cohen and Reshotko was in good agreement with experimental results but the application of this technique for other sweepback angles gave poor results.

7. The stagnation line heat transfer for various sweepback angles is reasonably predicted by the $\cos^{3/2}\Lambda$ law. The present experimental results are considerably lower than comparable data at low Mach numbers.

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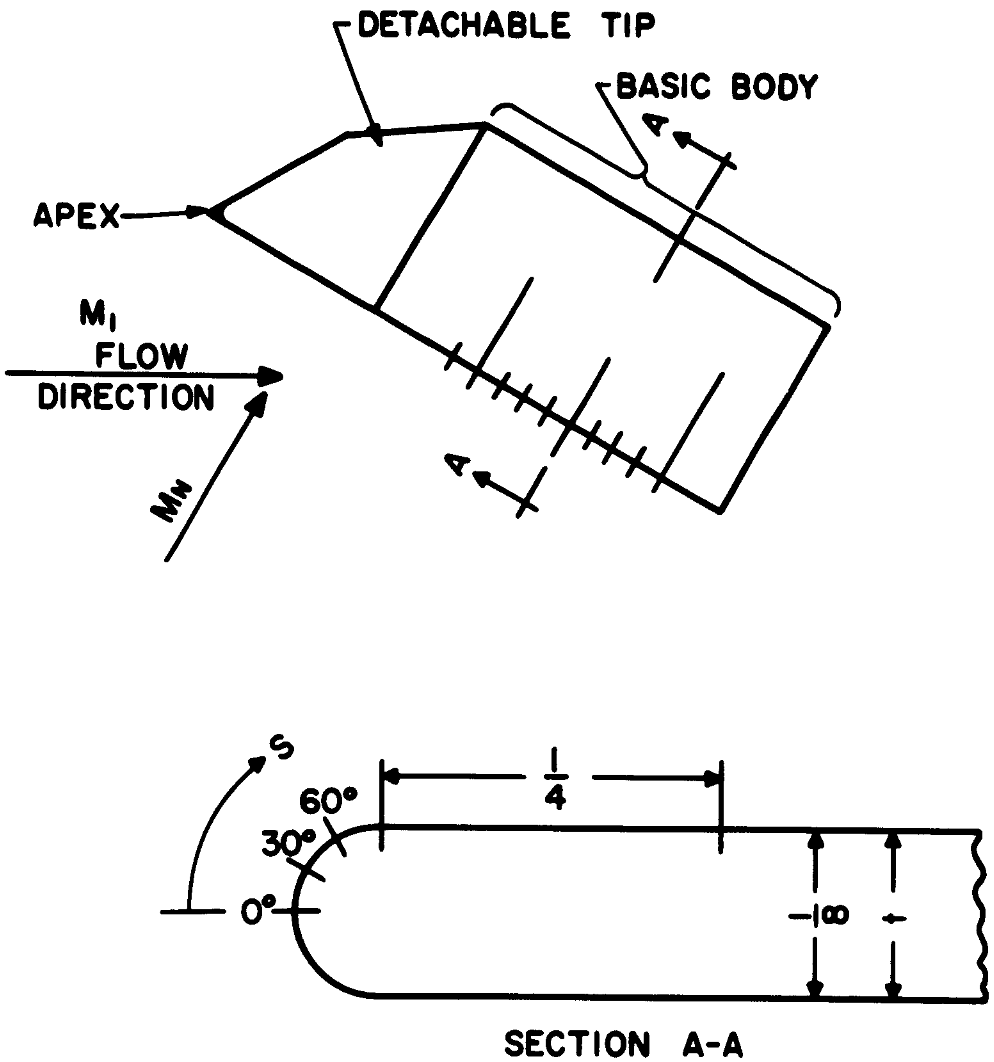
Figure 1. The heat transfer (left) and pressure (right) models with tips.



POINTS ARE PLACED 1/8 INCH APART ON STAGNATION LINE
AND 1/2 INCH APART ELSEWHERE

Figure 2. Location of the pressure orifices and thermocouples.

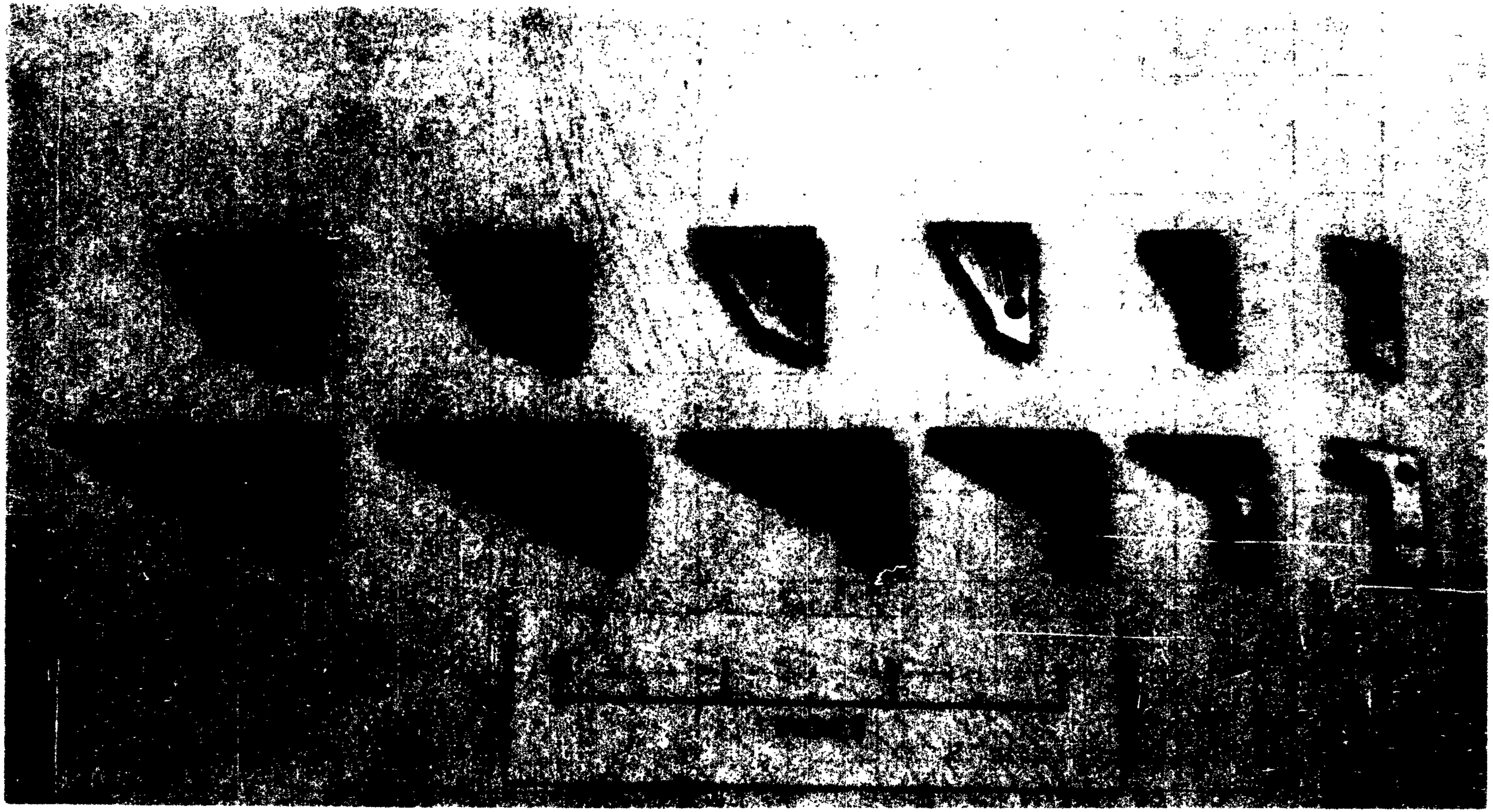


Figure 3. Various tips for the pressure model, $\Lambda = 75^{\circ}$ and $\Lambda = 60^{\circ}$.

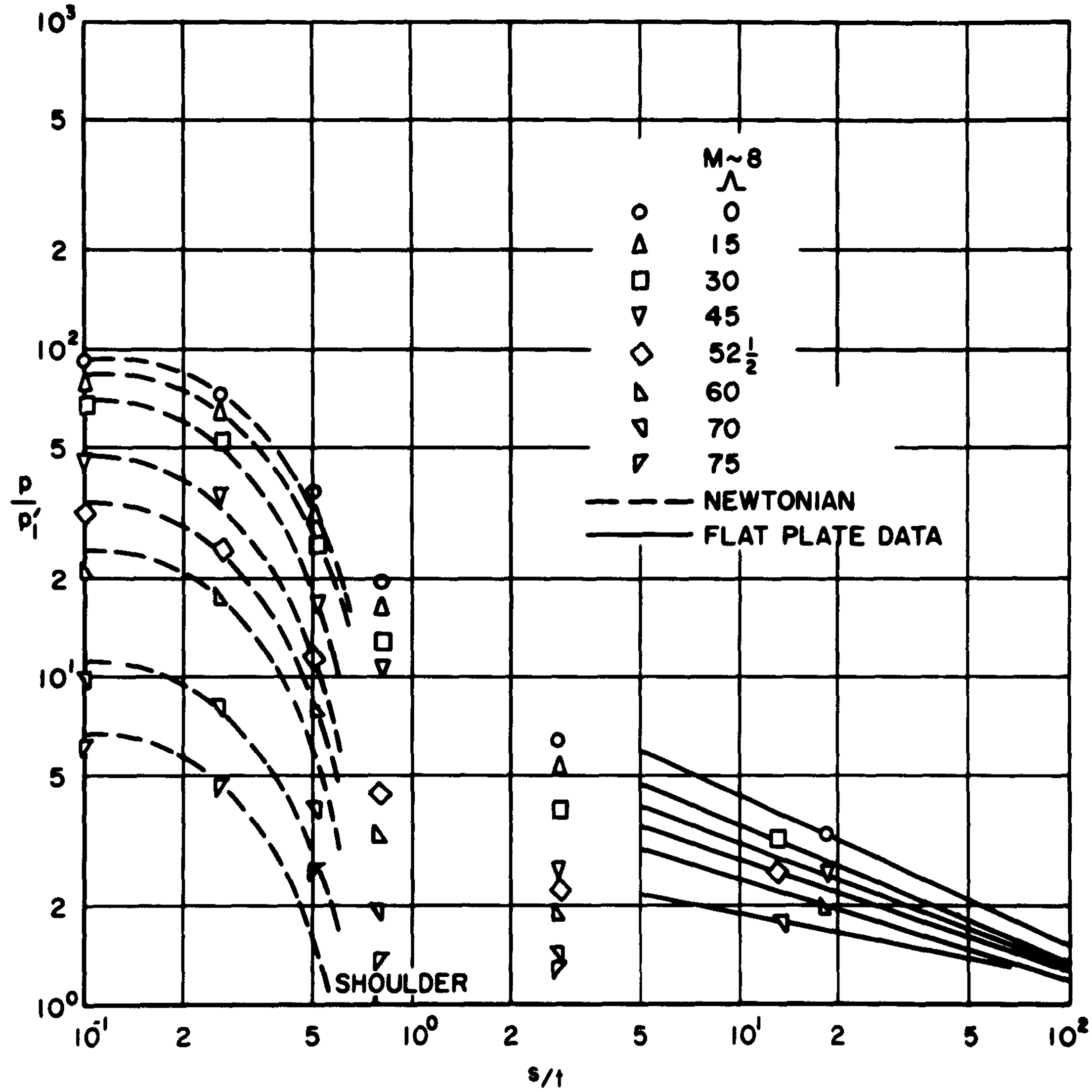


Figure 4. Summary of the pressure distributions at M~8.

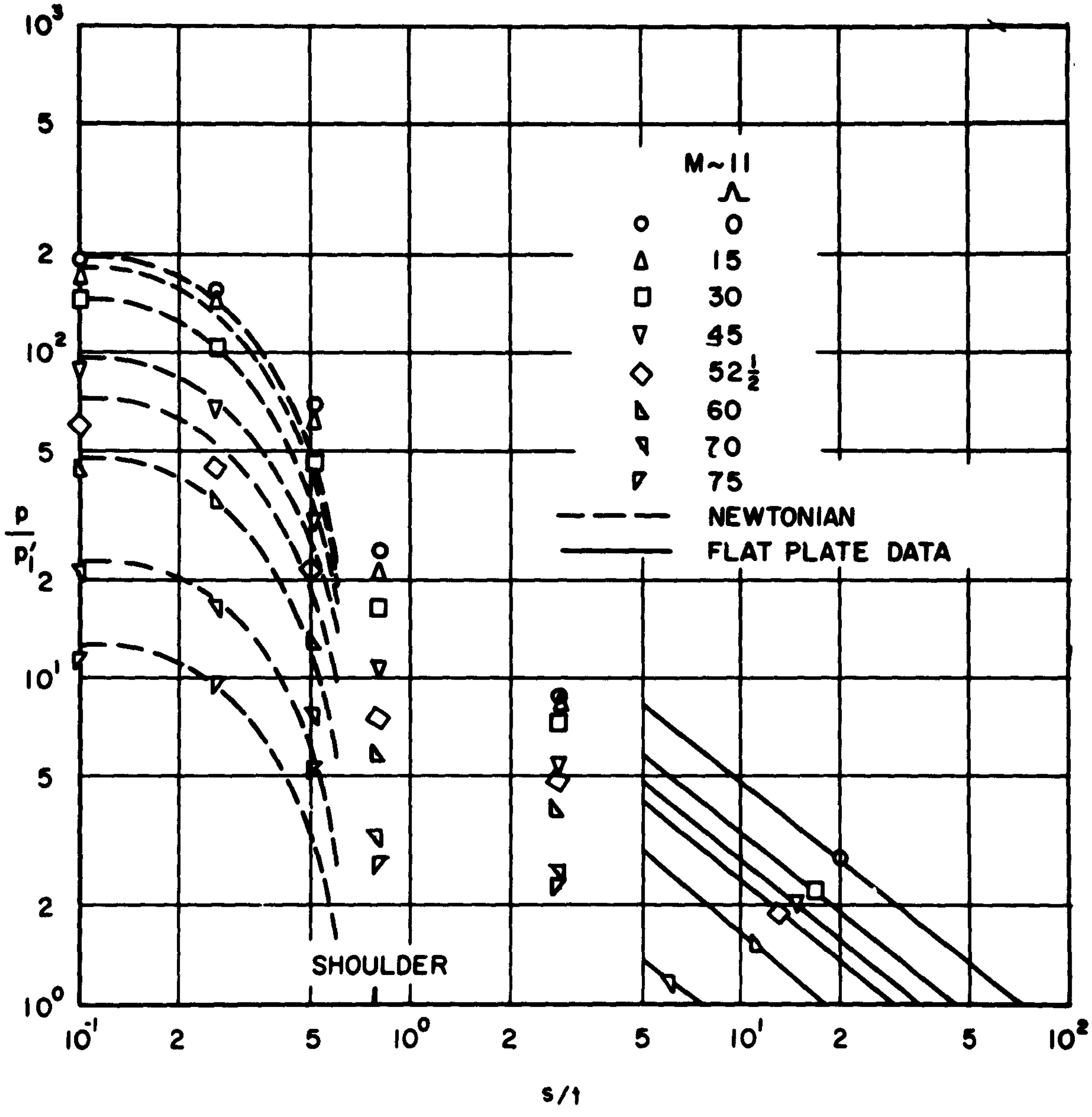


Figure 5. Summary of the pressure distributions at M∼11.

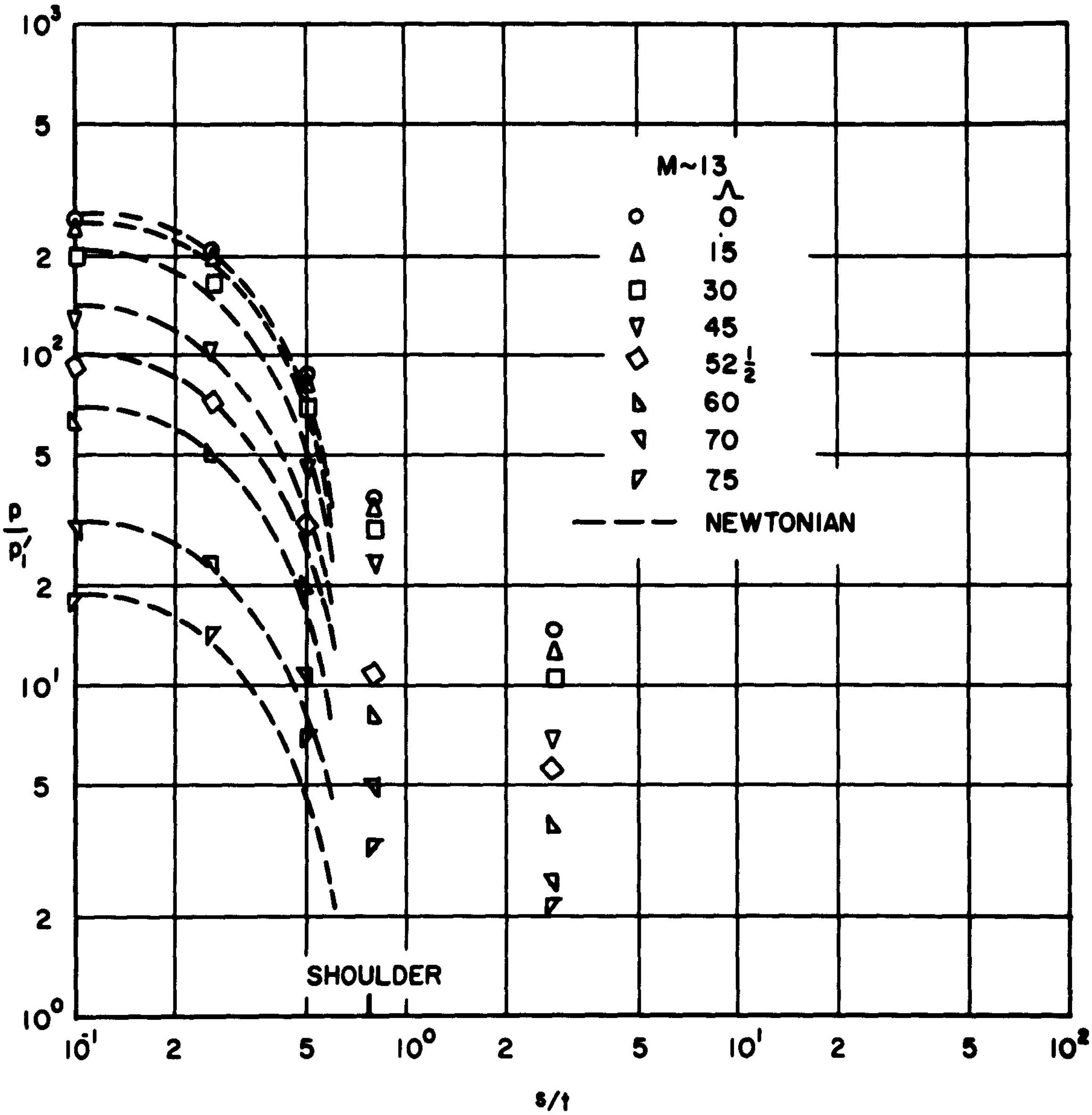


Figure 6. Summary of the pressure distributions at M∼13.

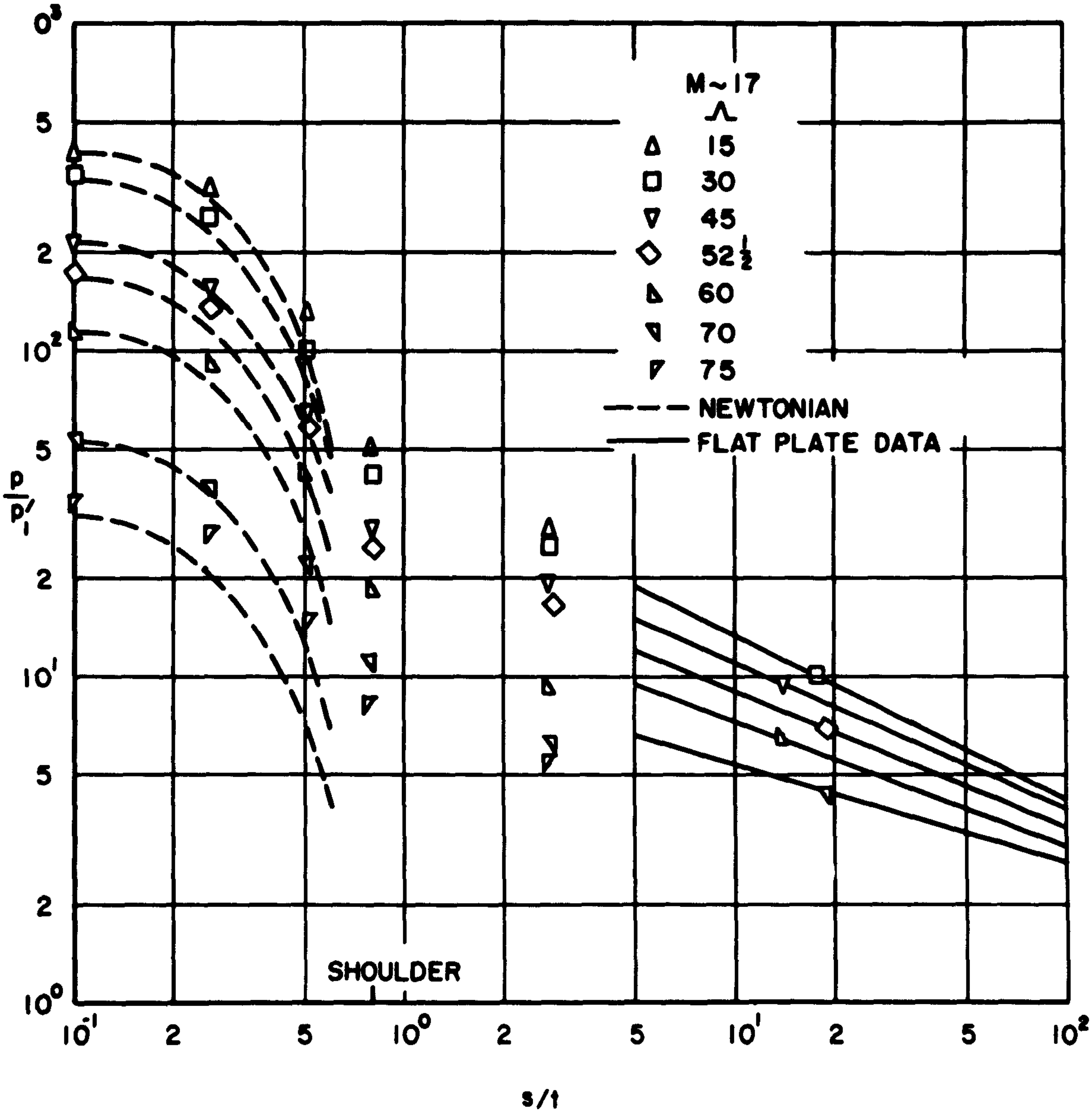


Figure 7. Summary of the pressure distributions at M~17.

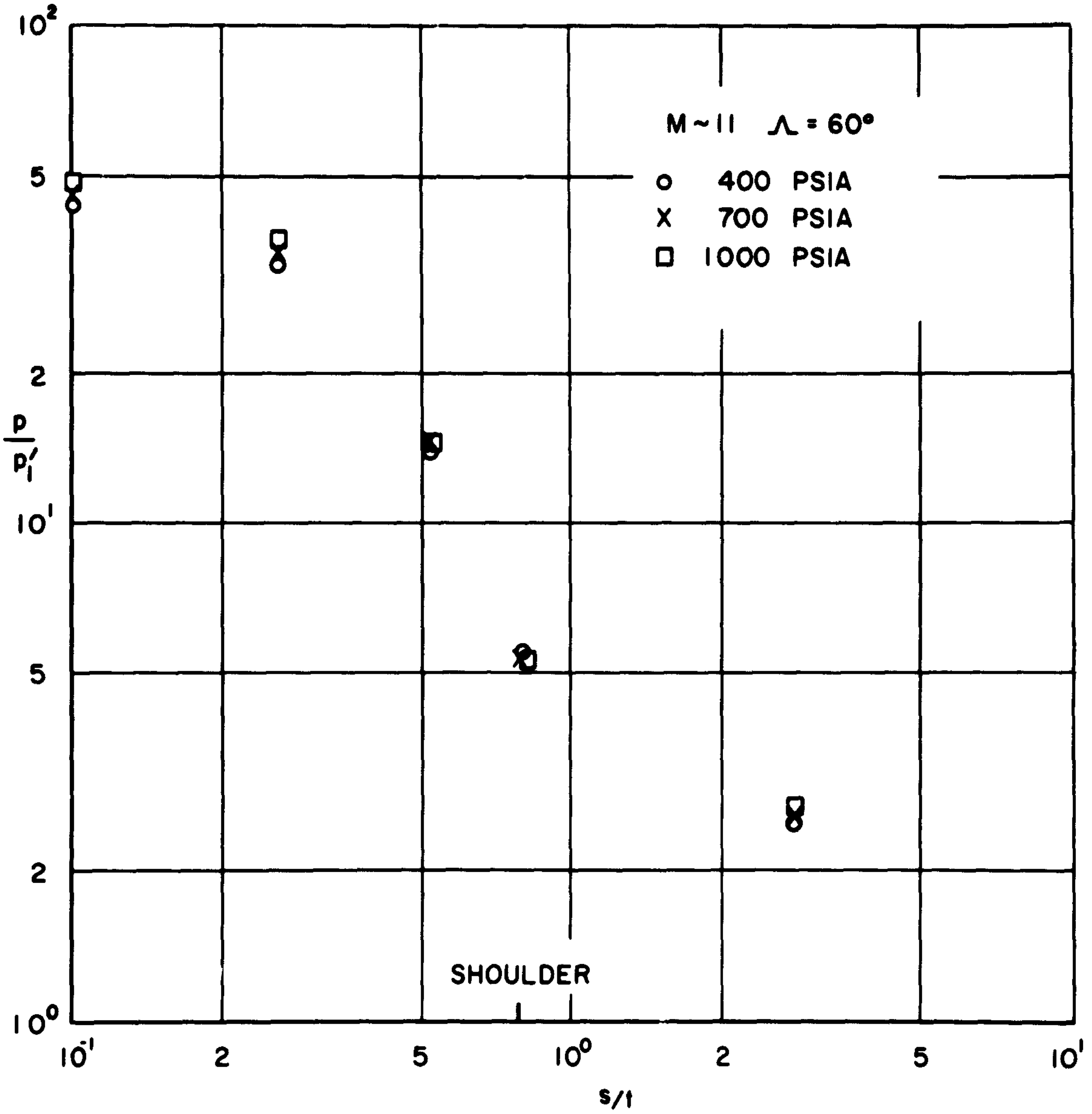


Figure 8. Effect of stagnation pressure at M~11, Λ= 60°.

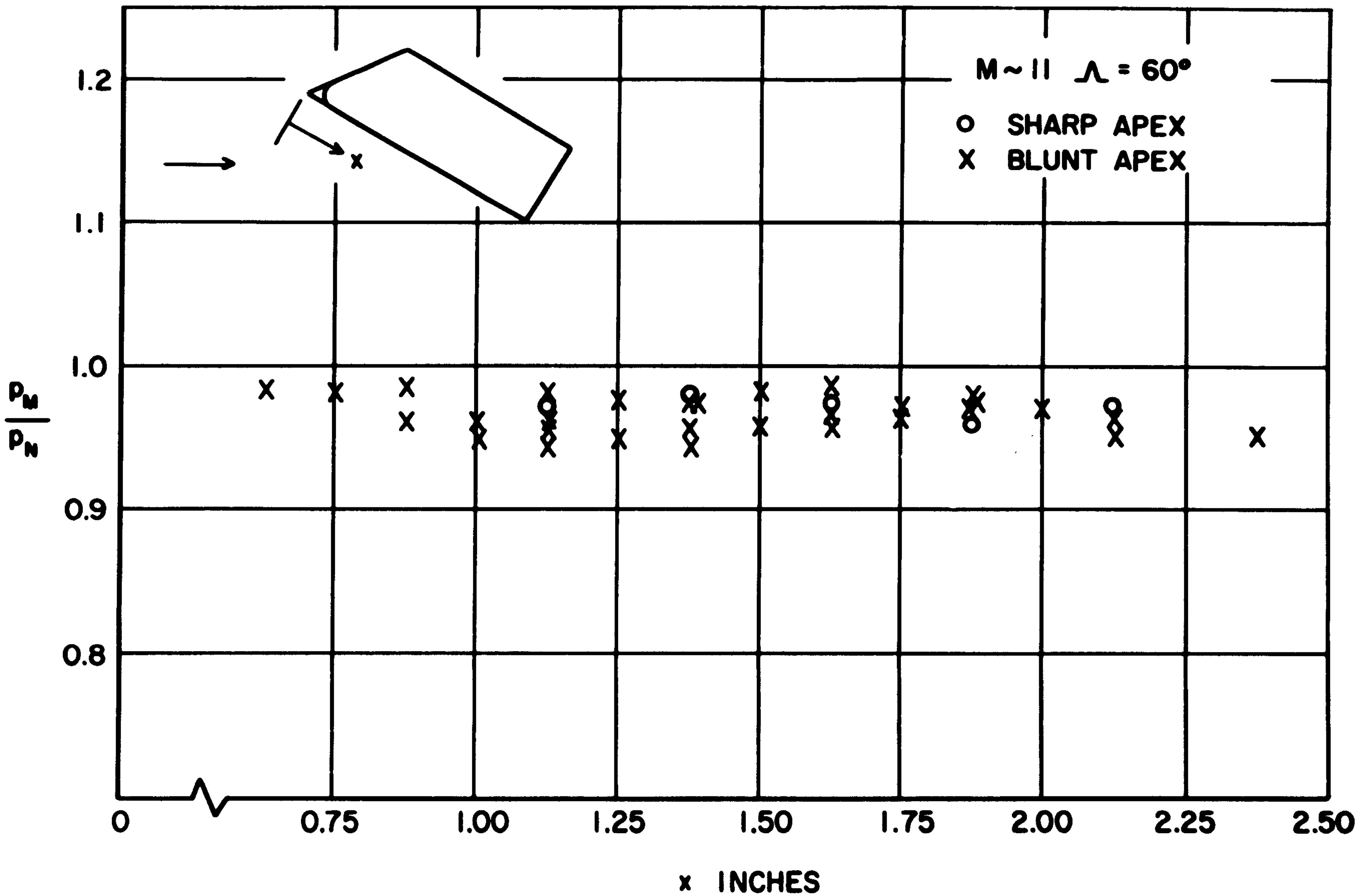


Figure 9. Normalized pressure ratio along the stagnation line for M~11, $\Lambda = 60^{\circ}$.

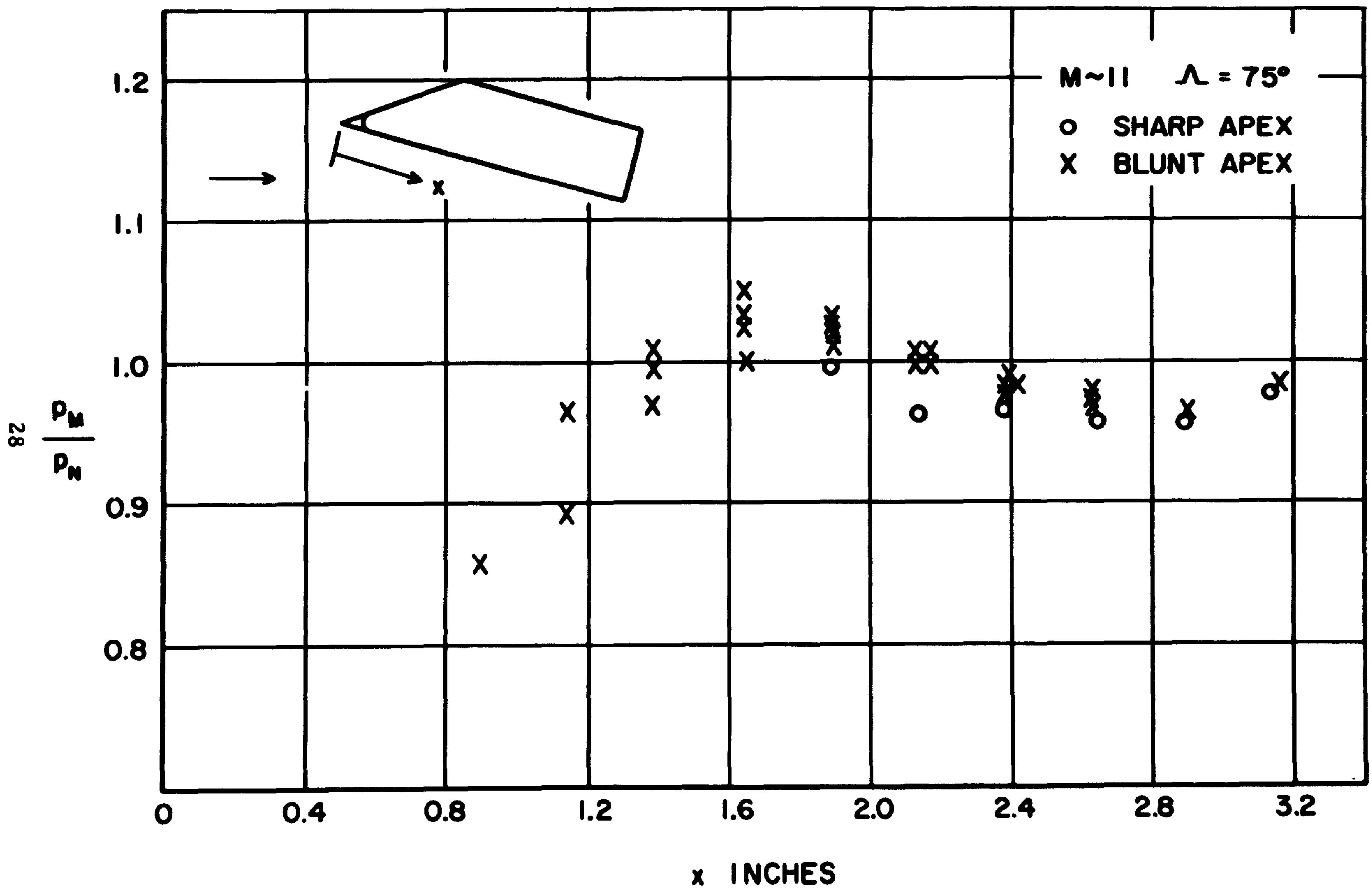


Figure 10. Normalized pressure ratio along the stagnation line for M~11, $\Lambda = 75^{\circ}$.

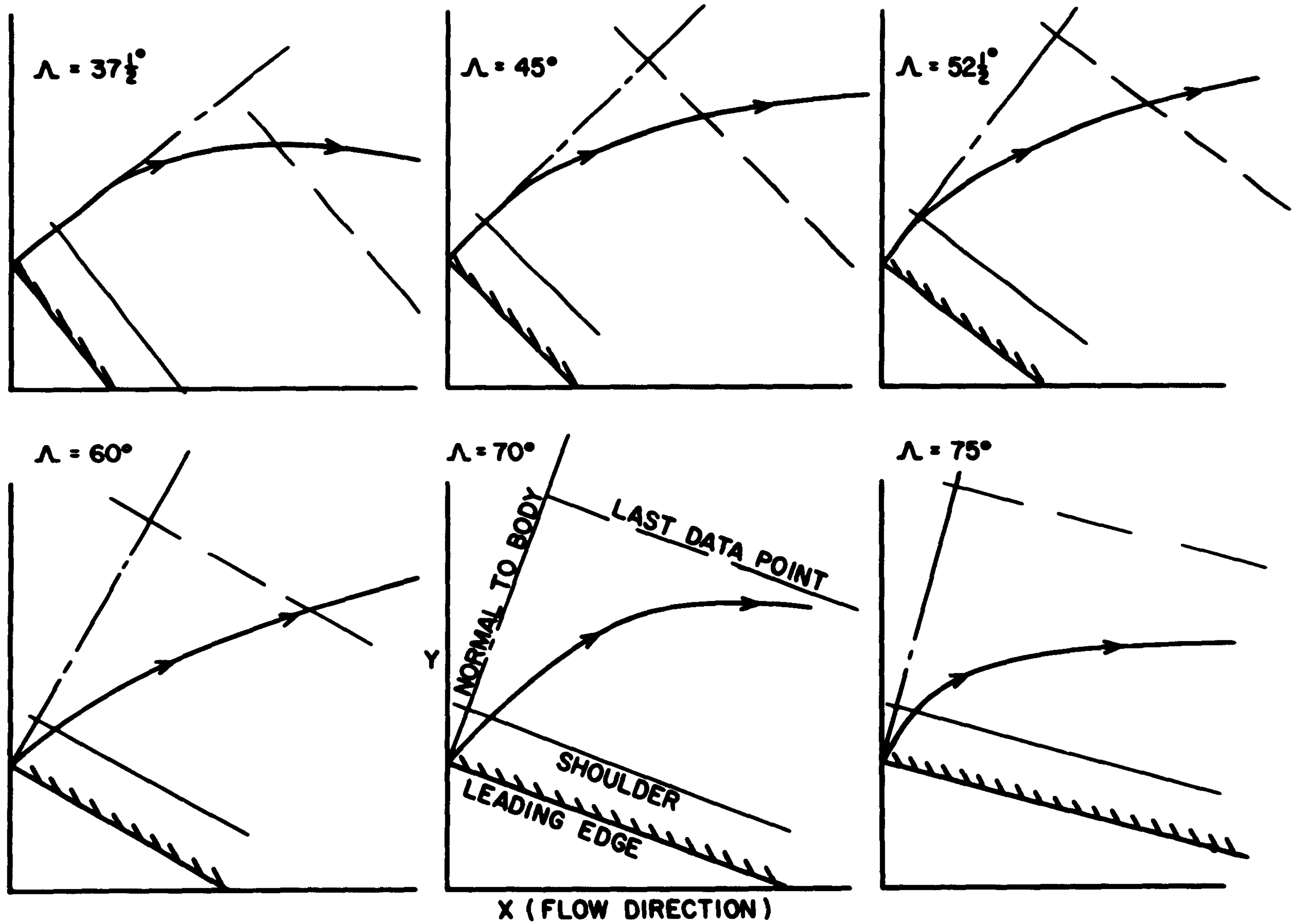


Figure 11. Summary of the oil trace studies for variable sweep at M~11.

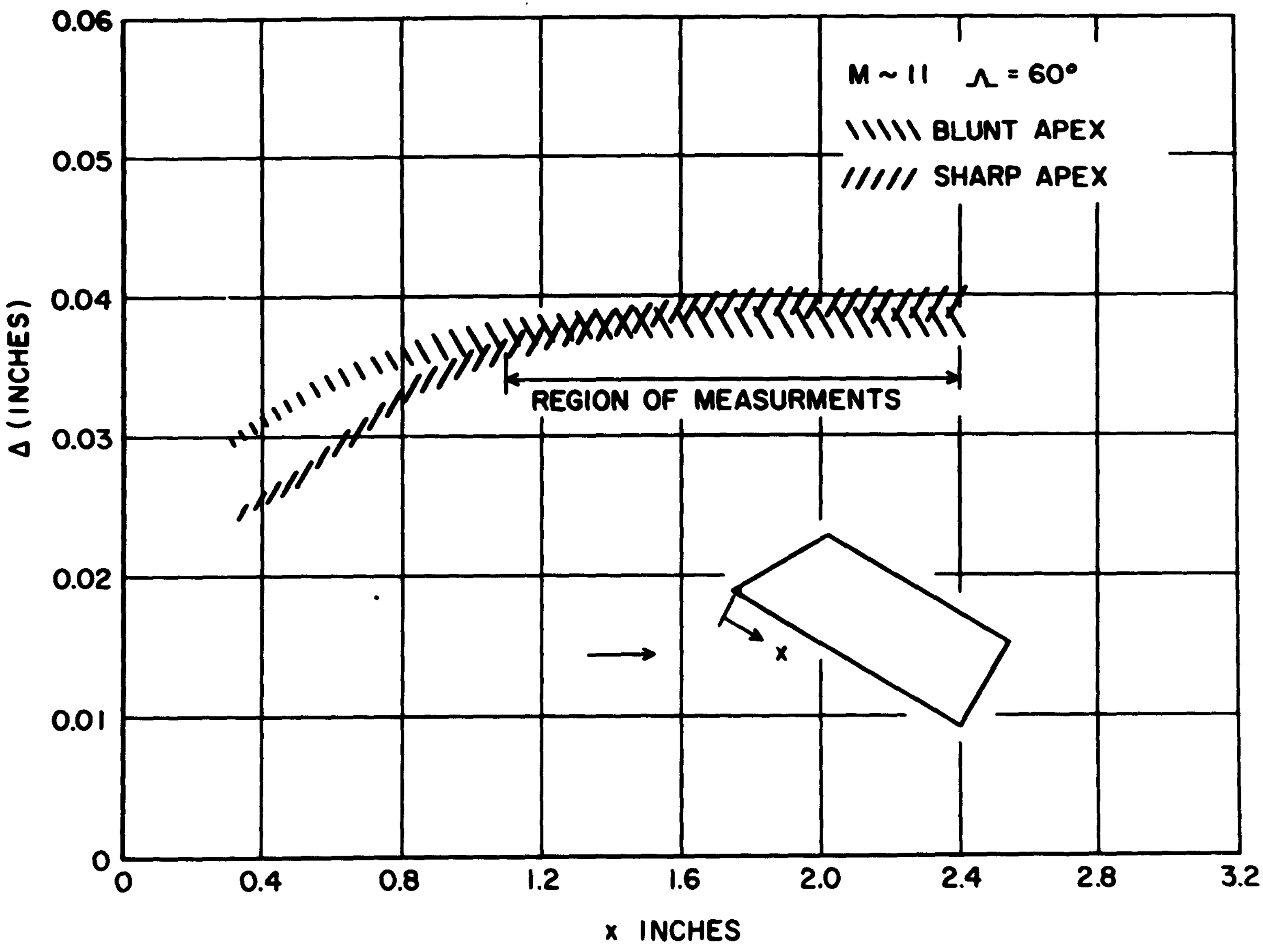


Figure 12. Stand-off distance for $M \sim 11$, $\Lambda = 60^{\circ}$.

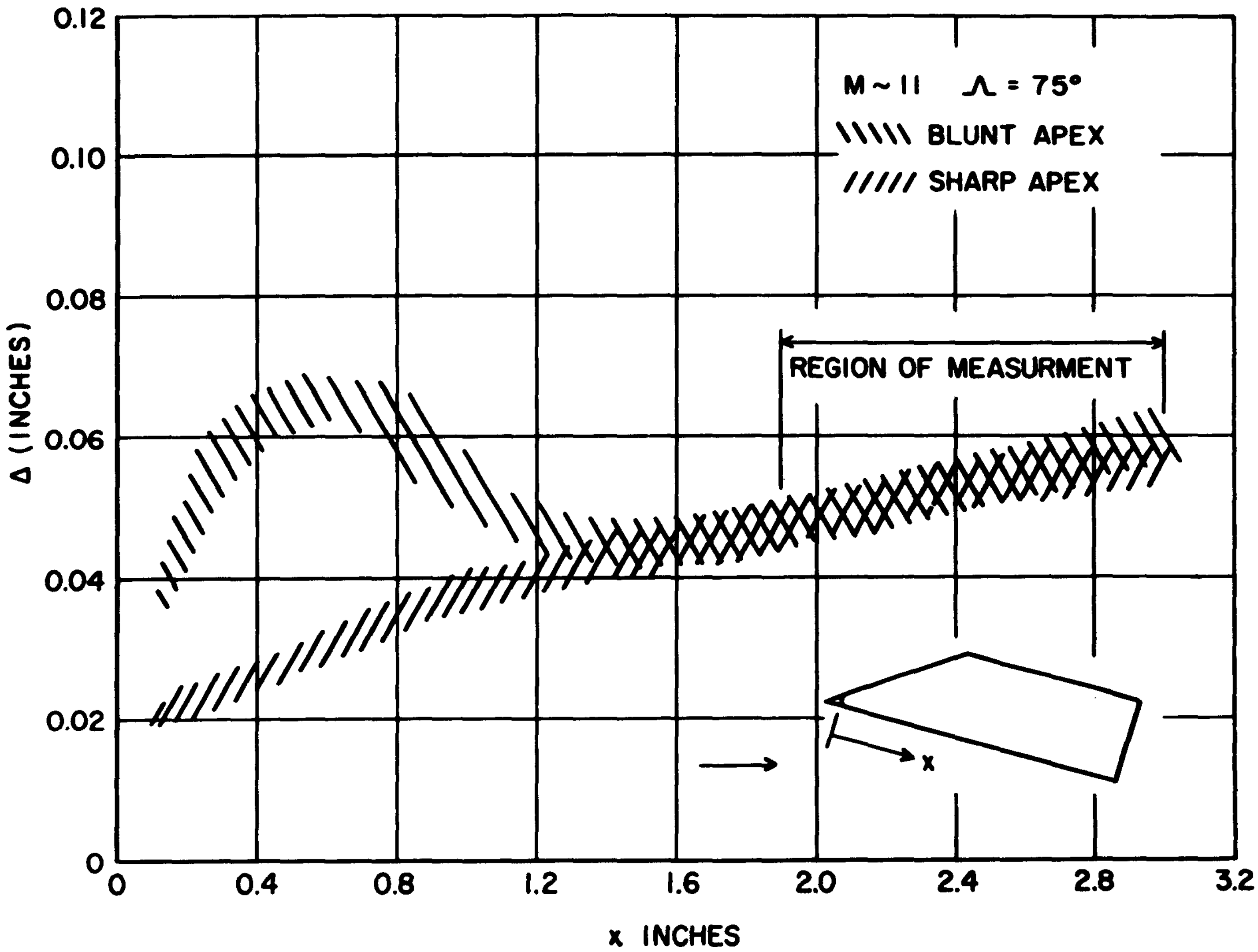


Figure 13. Stand-off distance for M∼11, Λ = 75°.

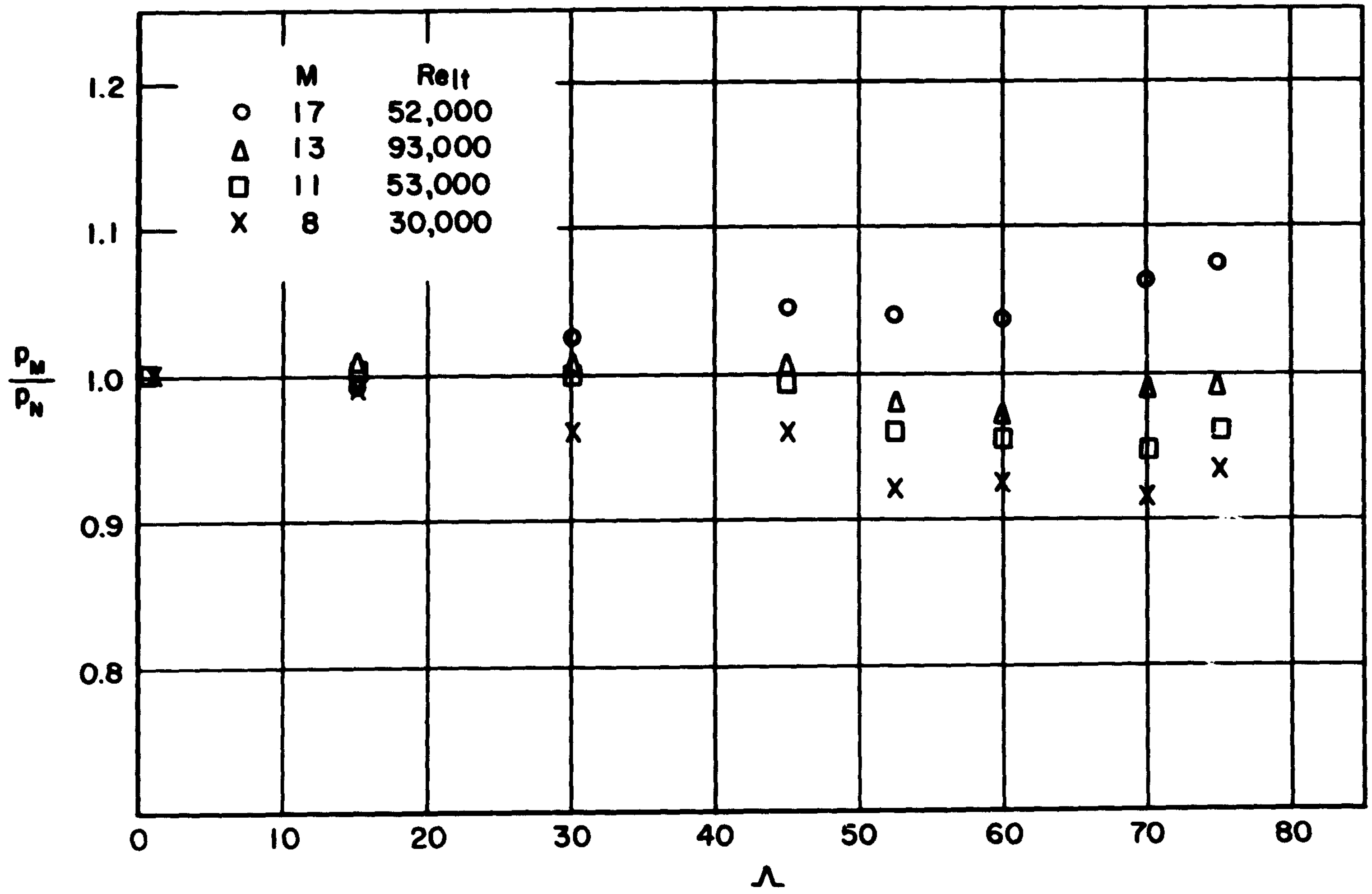


Figure 14. Effect of the normal Mach number hypothesis for various free stream Mach numbers.

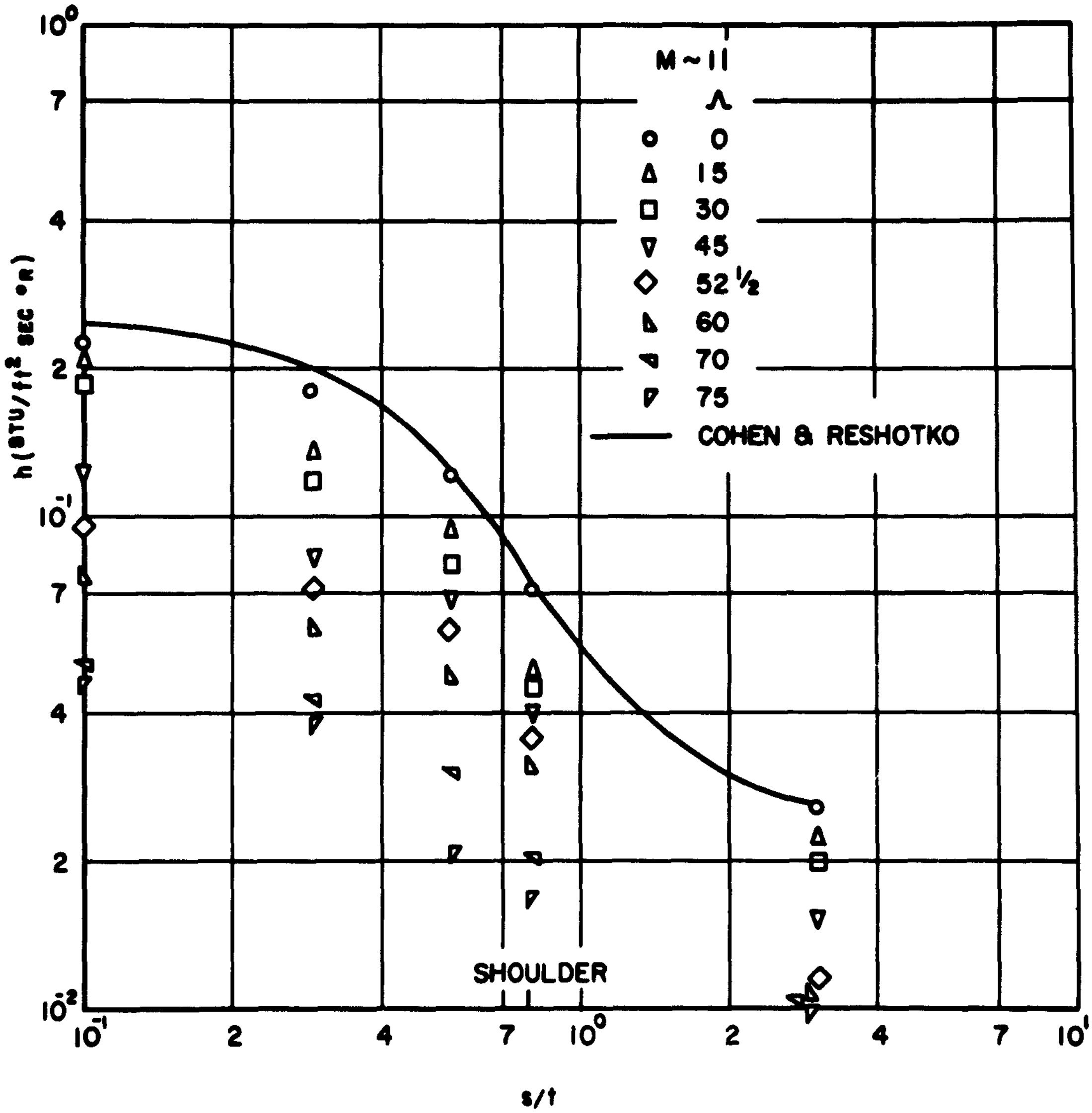


Figure 15. Summary of the heat transfer distribution at M~11.

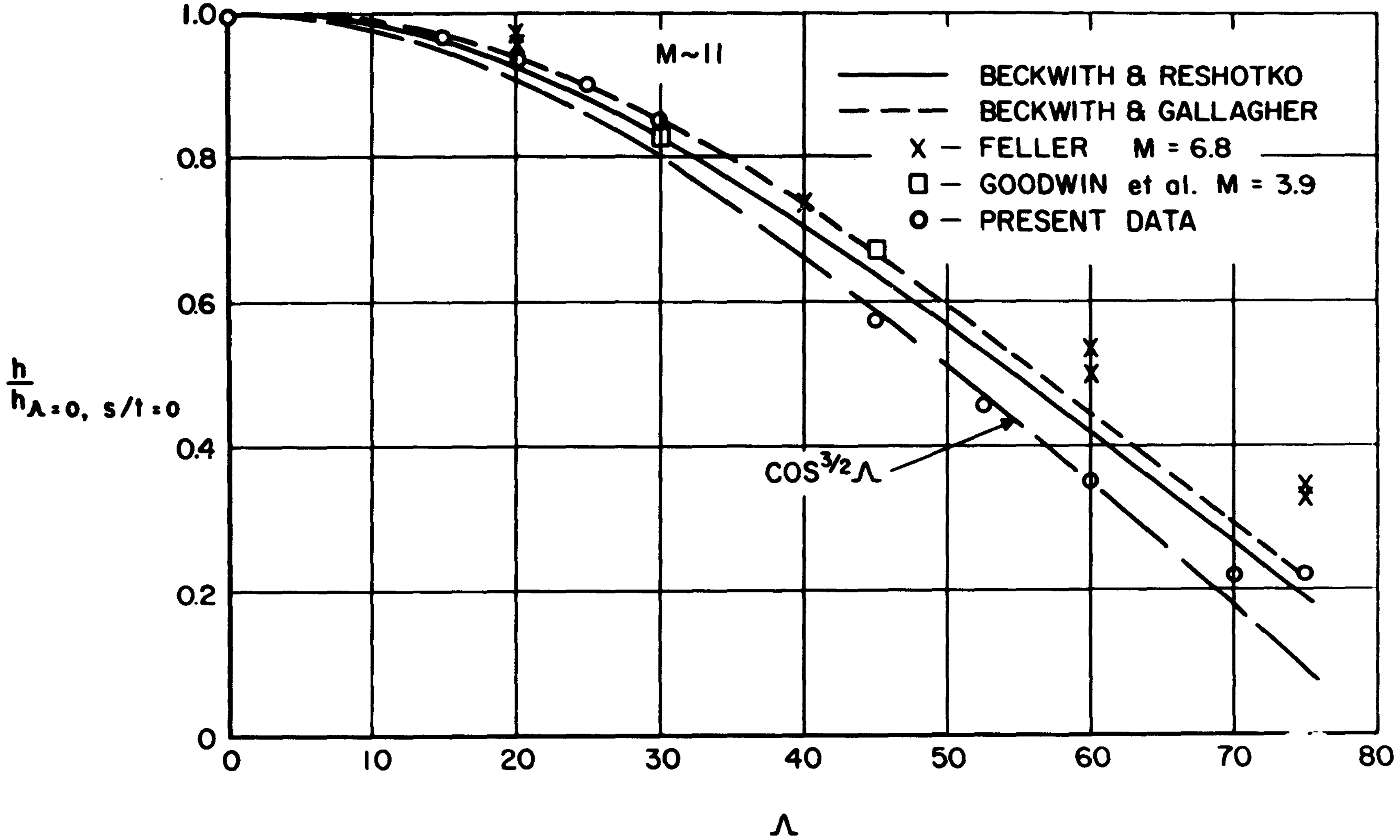


Figure 16. Effect of sweep on the heat transfer ratio at the stagnation line, M~11.

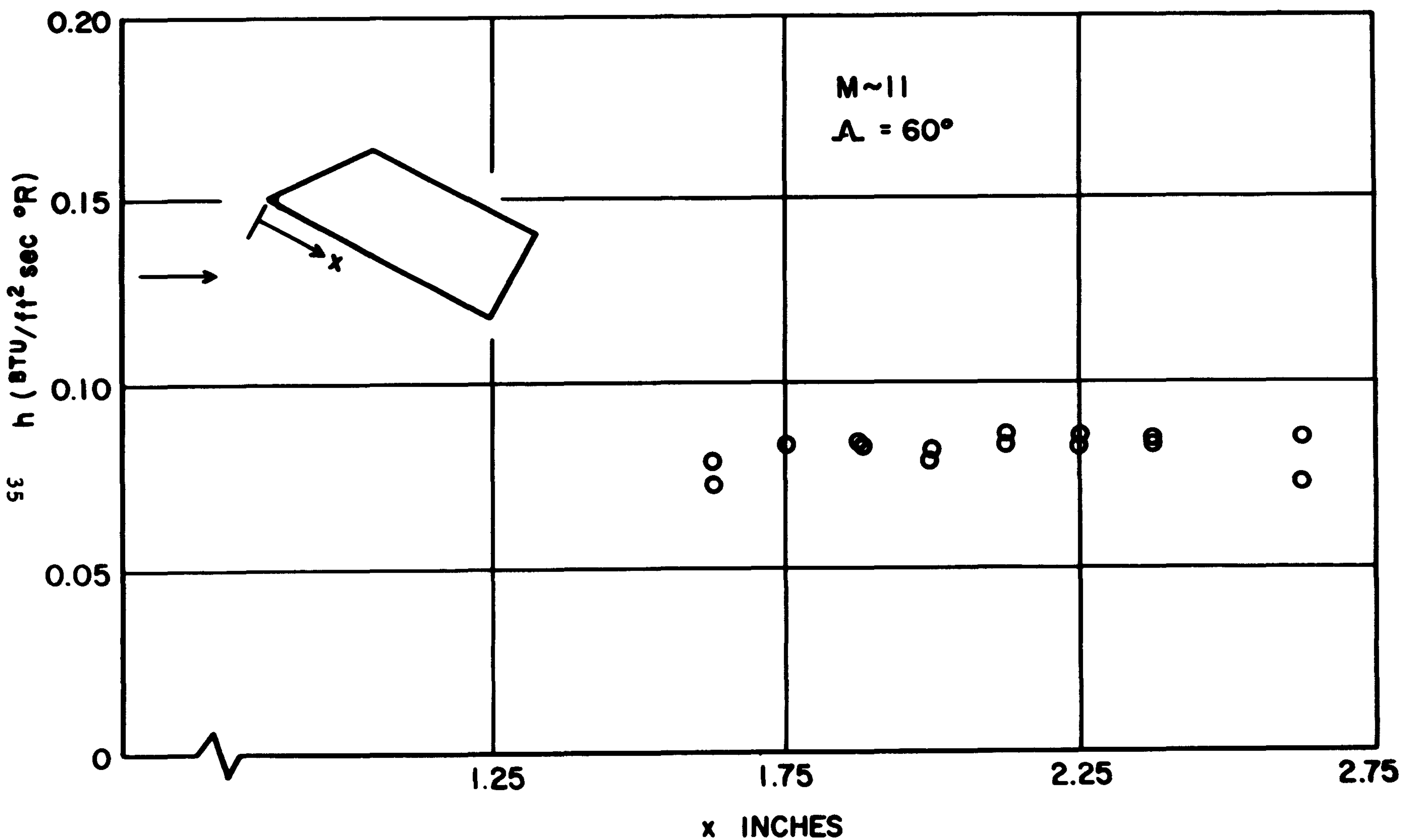


Figure 17. Stagnation line heat transfer at M~11, $\Lambda = 60^{\circ}$.

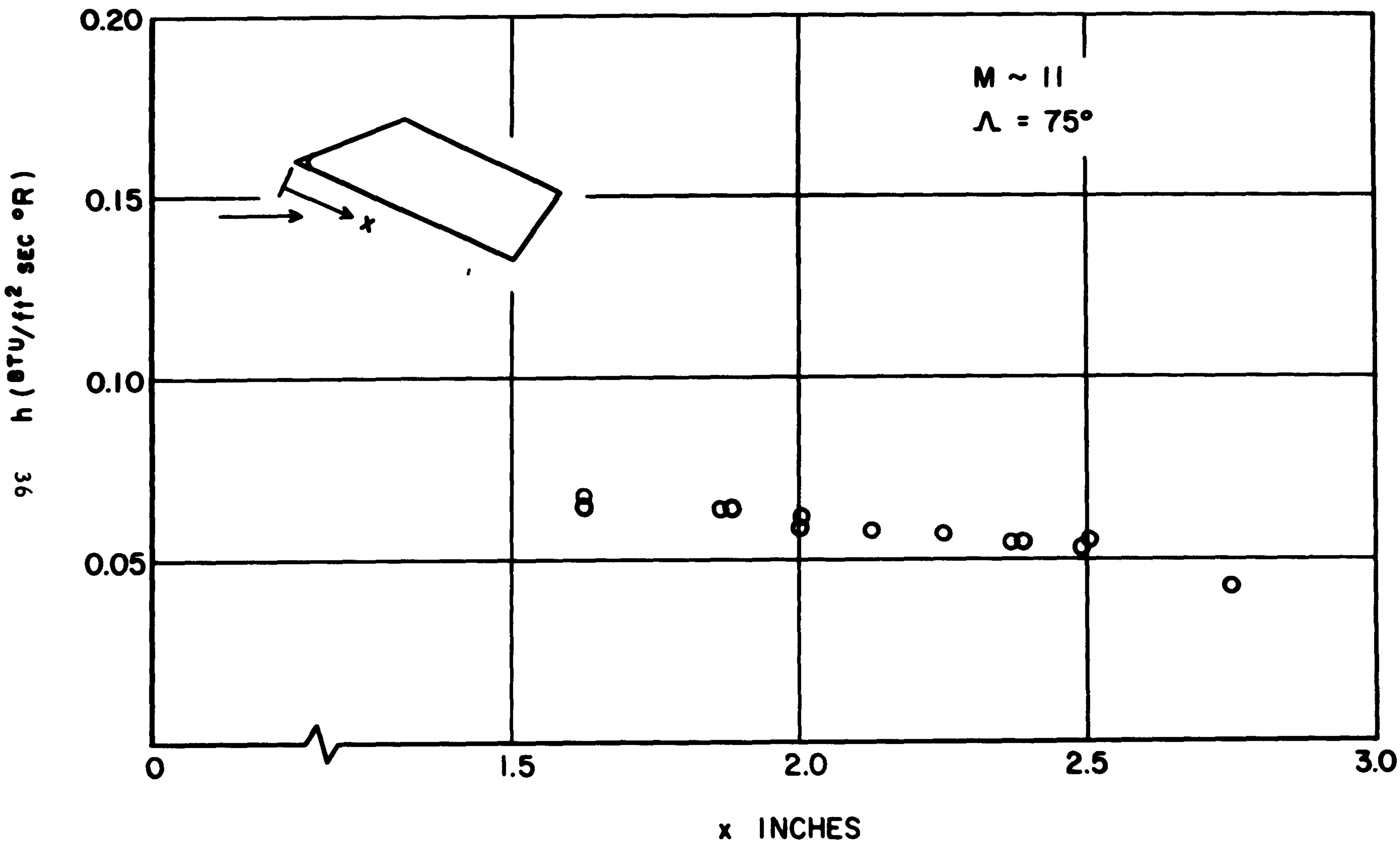


Figure 18. Stagnation line heat transfer at M~11, $\Lambda = 75^{o}$.

Aeronautical Research Laboratories, Wright-Patterson AFB, O. THE FLOW ABOUT THE LEADING EDGE OF A SWEPT BLUNT PLATE AT HYPERSONIC SPEEDS by C. C. Horstman, I. E. Vas, Princeton U., Princeton, N. J. August 1962. 36 p. incl. illus. (Project 7064; Task 7064-01) (Contract AF 33(616)-7629) (ARL 62-405)

Unclassified Report

As a part of a fundamental study of hypersonic wings, some detailed pressure distribution and heat transfer results have been obtained over sections of the leading edge region of a blunt plate at swept angles from

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zero to 75°. The tests were carried out in the Princeton University Helium Hypersonic Wind Tunnel at Mach numbers from 7 to 17. Some effects of the apex or upstream boundary of the plate on the leading edge regions under study were determined at various stations along the leading edge. The leading edge regions examined showed deviations from normal Mach number considerations at high sweep angles over the entire Mach number range studied.

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